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RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE CHARACTERISTICS OF THE J47-25

TURBOJET ENGINE - DATA PRESENTATION

By Paul E. Renas and Emmert T. Jansen

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE CHARACTERISTICS OF THE J47-25

TURBOJET ENGINE - DATA PRESENTATION

By Paul E. Renas and Emmert T. Jansen

SUMMARY

An investigation was conducted in an altitude test chamber at the NACA Lewis laboratory to determine the altitude performance of the J47-25 turbojet engine operating with a fixed-area exhaust nozzle. Data were obtained over a range of engine-inlet Reynolds numbers corresponding to altitudes from 18,000 to 54,000 feet and flight Mach numbers from 0.50 to 1.10.

Reducing the engine-inlet Reynolds number resulted in a reduction in corrected air flow but had essentially no effect on corrected exhaust-gas total temperature, corrected fuel flow, and engine pumping characteristics for a range of Reynolds number indices from 0.80 to 0.30. The corrected jet thrust parameter generalized throughout the range of engine-inlet Reynolds numbers investigated.

At a given corrected engine speed with critical pressure ratio existing in the exhaust nozzle, increasing the engine-inlet ram-pressure ratio from 1.0 to 1.25 decreased the corrected exhaust-gas temperature. Further increases in ram-pressure ratio had no effect on the exhaust-gas temperature.

INTRODUCTION

An investigation was conducted in an NACA Lewis altitude chamber to determine the altitude performance characteristics of a J47-25 axial-flow turbojet engine over a range of engine-inlet Reynolds number indices corresponding to altitudes from 18,000 to 54,000 feet and flight Mach numbers from 0.50 to 1.10. In order to simplify the procedure in obtaining performance data and to make the data applicable to any flight

condition, Reynolds number index $\frac{\delta_1}{\phi_1 \sqrt{\theta_1}}$, which is proportional to Reynolds number at a given corrected engine speed and is a function only

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of engine-inlet total pressure and temperature, was used instead of various set altitudes and flight Mach number combinations (reference 1). By the technique just mentioned, the data obtained in this investigation may be used to obtain the performance of the engine at any flight condition for which critical flow exists in the exhaust nozzle. An example is included in the appendix to illustrate the method of obtaining conventional performance parameters for a given flight condition from the data such as presented herein.

In addition to the basic engine performance, data were obtained in which the effects of variation of engine-inlet conditions on exhaust-gas temperature and thrust were observed. These effects are of importance from the standpoint of aircraft take-off and day-to-day weather variations.

All performance data obtained in this investigation are presented in both graphical and tabular form.

APPARATUS

Engine

The J47-25 axial-flow turbojet engine used in this investigation has a twelve-stage compressor, eight tubular combustion chambers, and a single-stage turbine. The engine has a static sea-level thrust rating of 6060 pounds at the rated engine speed of 7950 rpm and an engine manufacturer's turbine-outlet temperature of 1245° F. The compressor air flow is approximately 104 pounds per second and compressor pressure ratio is 5.3 at rated sea-level conditions. A conical exhaust nozzle having an area of 298.5 square inches was installed on the engine. Operation of the engine with this nozzle produced an average tail-pipe total gas temperature of 1710° R (1250° F), which is based on NACA instrumentation at static sea-level conditions and rated engine speed of 7950 rpm. The maximum dimensions of the engine are a 37-inch diameter and a 144-inch over-all length excluding the cylindrical tail pipe and the exhaust nozzle. The total weight of the engine is 2653 pounds.

Installation

The altitude test chamber in which the engine was installed is 10 feet in diameter and 60 feet in length. The test chamber is divided into three sections separated by bulkheads: the air-inlet section, the engine compartment, and the exhaust section. The engine was mounted on a thrust-measuring bed. A front bulkhead, which incorporated a labyrinth seal around the forward end of the engine, provided for freedom of movement of the engine in an axial direction. A rear bulkhead was installed to act as a radiation shield and to prevent recirculation of the hot exhaust gases about the engine.

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Instrumentation

The location of the instrumentation stations before and after each of the principal components of the engine is shown in figure 1. Sketches showing the arrangement of the separate temperature and pressure probes within a given station are presented in figure 2. The total-pressure tubes at stations 1 and 9 were located at the centers of 24 and 6 equal areas, respectively. The thermocouples at stations 1, 3, 5, and 9 and the total-pressure tubes at stations 3 and 5 were located on approximately equal spacings. The instrumentation at the engine inlet (station 1) was used in calculating the altitude and flight Mach number correction factors θ , δ , and ϕ . (All symbols are defined in the appendix.) The pressure and temperature measurements at station 9 were used to calculate ideal or rake jet thrust and nozzle gas flow. Measured jet thrust was also determined from scale readings for each condition investigated. The atmospheric pressure surrounding the jet nozzle was measured by four lip static probes located in the exhaust portion of the chamber (station 0).

Fuel flow was measured by two rotameters connected in series and calibration of the rotameters was made with the type fuel used in this investigation (MIL-F-5624A, grade JP-4).

PROCEDURE

The inlet conditions were varied to correspond to Reynolds number indices from 0.15 to 0.80. For each inlet condition, the exhaust pressure was reduced to the minimum of the exhaust system with the engine operating at rated speed. The inlet temperature and pressure and the exhaust pressure were then maintained constant while data were taken over a range of engine speeds from rated speed to approximately the speed where the exhaust nozzle became unchoked. A summary of the operating conditions covered in this investigation is given in the following table:

Reynolds number index	Inlet total temperature (°R)	Inlet total pressure (lb/sq ft)	Ram- pressure ratio
0.15	410	232	1.19
.2	410	315	1.48
.25	410	387	1.64
.3	410	465	1.34
.3	410	465	1.70
.4	467	739	1.35
.425	437	718	1.41
.5	467	923	1.95
.6	467	1108	2.14
.8	530	1740	1.70

The methods of calculation are given in the appendix.

PRESENTATION OF DATA

The simulated altitude performance data obtained in this investigation were corrected to NACA standard altitude conditions and are presented in table I. Generalization of data for various engine-inlet conditions corresponding to a given Reynolds number index requires that critical flow be established in the exhaust nozzle. The range of corrected engine speeds over which the exhaust nozzle of the engine was choked is shown in figure 3 for a range of Reynolds number indices corresponding to various altitudes and flight Mach numbers. At all altitudes, this minimum corrected engine speed at which choking occurred decreased approximately linearly from about 7600 rpm at a flight Mach number of 0.2 to about 5750 rpm at a flight Mach number of 1.10. The data of this report may be used to determine performance only at flight conditions in the choked region above this curve.

In order to aid in determining the Reynolds number index corresponding to a given flight condition and thereby determine the engine performance at NACA standard altitude conditions from the generalized data presented, the values of δ , θ , ϕ , and Reynolds number index are given in table II for a wide range of flight conditions; 100 percent ram-pressure recovery was assumed.

Effect of Engine-Inlet Conditions on Performance

In addition to the basic engine performance, two effects of special concern regarding exhaust-nozzle sizing and aircraft take-off are the effect of engine-inlet temperature on exhaust-gas temperature at sea-level static-pressure conditions and the effect of engine-inlet ram-pressure ratio on exhaust-gas temperature and thrust at low flight speeds and low altitudes. However, because of test-facility limitations, these effects had to be investigated at altitudes of 15,000 and 20,000 feet, respectively.

The effect of engine-inlet total temperature on exhaust-gas total temperature is presented in figure 4 for a constant actual engine speed of 7950 rpm. A decrease in inlet total temperature from 532° R to 465° R resulted in a decrease in exhaust-gas total temperature of approximately 50° R, and any further decrease in inlet temperature caused the exhaust-gas temperature to increase. The data for the performance variables presented in figure 4 along with other engine performance data are included in table III.

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The effect of engine-inlet ram-pressure ratio on corrected exhaust-gas total temperature and the corresponding net thrust variation for various corrected engine speeds are shown in figure 5. The decrease in corrected exhaust-gas total temperature as ram-pressure ratio is increased results from an increase in effective flow area of the exhaust nozzle, which corresponds to an increase in nozzle flow coefficient. The change in effective flow area is caused by the fact that the exhaust nozzle is not fully choked and by the existence of a boundary layer of subsonic flow around the sonic jet. This layer of subsonic flow decreases in depth as the engine-inlet ram-pressure ratio is increased, thus increasing the effective area of the nozzle and reducing the tail-pipe temperature. The effect of this flow-area change becomes constant after a ram-pressure ratio of approximately 1.25 (which corresponds to a tail-pipe pressure ratio of approximately 2.5) is attained. At this ram-pressure ratio of 1.25, the net thrust loss is approximately 3 percent of the thrust that could be obtained if the exhaust-gas total temperature had remained constant at the value obtained for an engine-inlet static condition. A tabulation of these data along with other engine performance parameters is given in table IV.

General Performance Calibration Data

The effect of Reynolds number index on generalized engine performance is shown in figures 6 to 10 where the corrected air flow, corrected fuel flow, corrected jet thrust parameter, corrected exhaust-gas total temperature, and engine pumping characteristics are presented. The variation of corrected air flow with corrected engine speed for various Reynolds number indices is presented in figure 6. At a corrected engine speed of 7950 rpm, the corrected air flow decreased from 104.0 to 99.2 pounds per second as Reynolds number index was decreased from 0.80 to 0.15. The corrected fuel flow (fig. 7) generalized for Reynolds number indices from 0.80 to 0.30 at corrected engine speeds above about 7500 rpm but increased with a further reduction of Reynolds number index. This increase in fuel flow results from the required rise in turbine-inlet temperature due to the decrease in compressor efficiency and the decrease in combustion efficiency at low Reynolds number indices. The increase in corrected fuel flow at rated corrected engine speed was approximately 8 percent as Reynolds number index was reduced from 0.30 to 0.15. The corrected jet thrust parameter, based on scale thrust readings, (fig. 8) generalized throughout the range of Reynolds number indices and corrected engine speeds investigated. Corrected exhaust-gas total temperature (fig. 9) generalized for Reynolds number indices from 0.80 to 0.30 but increased with a further reduction in Reynolds index. This increase in corrected exhaust-gas total temperature at the lower Reynolds numbers is attributed primarily to the decrease in compressor efficiency, which requires more work from the

turbine to maintain a given engine speed and hence a higher turbine-inlet temperature. Figure 10 illustrates the effect of Reynolds number index on the engine pumping characteristics. The relation between engine total-pressure ratio and engine total-temperature ratio is defined by a single line as Reynolds number index is decreased from 0.80 to 0.30 but shifts in the direction of increased engine total-temperature ratio at a given engine total-pressure ratio for a further reduction in Reynolds number index. This shift in the curves reflects the reduced efficiency of the compressor and turbine at conditions of low inlet Reynolds number.

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The corrected engine windmilling speed is shown in figure 11 as a function of flight Mach number for altitudes from 15,000 to 45,000 feet. The corrected engine windmilling speed was unaffected by changes in altitude for the range of flight Mach numbers investigated.

The thrust is dependent upon the exhaust-gas temperature and in this investigation the gas temperatures were measured by the engine manufacturer's four-probe and five-probe thermocouple harnesses as well as the 25 NACA thermocouples. The readings of these different sets of instrumentation differ, with the result that the thrust at a given measured temperature will also vary. A comparison of the thrusts obtained is presented in the following table for NACA standard sea-level static conditions:

Performance based on	Engine speed (rpm)	Engine manufacturer's exhaust-gas thermocouple reading $T_{9,1}$ ($^{\circ}$ R)	Exhaust-gas total temperature based on NACA instrumentation T_g ($^{\circ}$ R) (a)	Thrust (lb)
Exhaust-gas total temperature of 1710° R	7950	----	1710	5894
Engine manufacturer's five-probe thermocouple harness	7950	1710	1760	6074
Engine manufacturer's four-probe thermocouple harness	7950	1710	1766	6098

^aBased on an average of 25 NACA thermocouples located 15.15 in. downstream of tail-cone-outlet flange.

The exhaust nozzle (area, 298.5 sq in.) was sized so as to give an exhaust-gas temperature of 1710° R (1250° F) at standard sea-level static conditions and rated engine speed. For this exhaust-gas temperature of 1710° R, the standard sea-level static thrust is 5894 pounds.

Because the engine is normally rated by the manufacturer for an exhaust-gas temperature based on a thermocouple reading obtained from the four- or five-probe thermocouple harness, thrust values have been included in the preceding table for the thermocouple reading of 1710° R obtained from the four- and five-probe systems with the corresponding gas temperatures included. The four- and five-probe harnesses indicated an exhaust-gas temperature between 50° and 60° lower than the true gas temperature and therefore give a correspondingly higher thrust for a given temperature limit based on a thermocouple reading. The method employed in calculating the thrust values is given in the appendix.

SUMMARY OF RESULTS

The following results were obtained from an investigation of the altitude performance of a J47-25 turbojet engine in an altitude chamber over a range of engine-inlet Reynolds number indices from 0.15 to 0.80:

1. At a constant engine speed, a decrease in inlet total temperature from 532° to 465° R resulted in a decrease in exhaust-gas total temperature of approximately 50° R.
2. At a given corrected engine speed and with critical pressure ratio existing in the exhaust nozzle, the corrected exhaust-gas temperature decreased as the ram-pressure ratio was increased from 1.0 to 1.25. Further increases in ram-pressure ratio had no effect on temperature. The corresponding net thrust loss at ram-pressure ratios of 1.25 and above, due to the reduction in exhaust-gas temperature below the limiting value, amounted to 3 percent.
3. At a corrected engine speed of 7950 rpm, the corrected air flow decreased from 104.0 to 99.2 pounds per second as Reynolds number index was decreased from 0.80 to 0.15.
4. Corrected exhaust-gas total temperature, corrected fuel flow, and engine pumping characteristics generalized for Reynolds number indices from 0.80 to 0.30 and the corrected jet thrust parameter generalized throughout the range of Reynolds number indices and corrected engine speeds investigated.
5. The corrected engine windmilling speed was unaffected by changes in altitude for the range of flight Mach numbers investigated.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 3, 1952

APPENDIX - METHODS OF CALCULATION

Symbols

The following symbols are used in the calculation and on the figures:

- A area, sq ft
C_T thermal expansion coefficient, ratio of hot exhaust-nozzle area to cold exhaust-nozzle area
C_d ratio of effective flow area to physical flow area
C_j jet thrust coefficient
F_d thrust system scale reading, lb
F_j jet thrust, lb
F_n net thrust, lb
f/a fuel-air ratio
g acceleration due to gravity, 32.2 ft/sec²
M Mach number
N engine speed, rpm
P total pressure, lb/sq ft absolute
p static pressure, lb/sq ft absolute
R gas constant, 53.3 ft-lb/(lb)(°R)
Re Reynolds number index, $\frac{\delta_1}{\Phi_1 \sqrt{\theta_1}}$
T total temperature, °R
T_i indicated total temperature, °R
V velocity, ft/sec
W_a air flow, lb/sec

W_f fuel flow, lb/hr

W_g gas flow, lb/sec

γ ratio of specific heats

δ ratio of engine-inlet total pressure P_1 to NACA standard sea-level pressure, 2116 lb/sq ft

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ϕ ratio of coefficient of viscosity corresponding with T_1 to coefficient of viscosity corresponding with NACA standard sea-level temperature, 519° R

Subscripts:

0 free-stream conditions

0a bellmouth inlet

1 engine inlet

2 compressor inlet

3 compressor outlet

5 turbine outlet

9 exhaust-nozzle inlet

10 exhaust-nozzle outlet

c1 compressor 12-stage leakage air flow

d thrust-cell measurement

e equivalent

i indicated

n vena contracta at exhaust-nozzle outlet

r rake

s scale

Calculations

Flight Mach number and velocity. - The flight Mach number assuming complete ram-pressure recovery was computed as

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1} \left[\left(\frac{P_1}{P_0} \right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1 \right]} \quad (1)$$

and

$$V_0 = M_0 \sqrt{\gamma_1 g R T_1 \left(\frac{P_0}{P_1} \right)^{\frac{\gamma_1 - 1}{\gamma_1}}} \quad (2)$$

Temperature. - Total temperature was determined by a calibrated thermocouple with an impact-recovery factor of 0.85 from the indicated temperature by the following equation:

$$T = \frac{T_i \left(\frac{P}{P_i} \right)^{\frac{\gamma - 1}{\gamma}}}{1 + 0.85 \left[\left(\frac{P}{P_i} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]} \quad (3)$$

Engine air flow. - Because of the large amount of air-flow leakage at the station where the engine inlet screens are mounted, the gas flow was determined at the exhaust-nozzle exit from total pressure and temperature at the nozzle inlet (station 9) by the following equation with the assumption that no energy loss occurred between the nozzle inlet and exit:

$$W_{g,n} = C_T C_d A_{10} p_n \sqrt{\frac{2 \gamma_9}{\gamma_9 - 1} \frac{g}{R T_9} \left[\left(\frac{P_9}{P_n} \right)^{\frac{\gamma_9 - 1}{\gamma_9}} - 1 \right] \left(\frac{P_9}{P_n} \right)^{\frac{\gamma_9 - 1}{\gamma_9}}} \quad (4)$$

where in the subsonic case

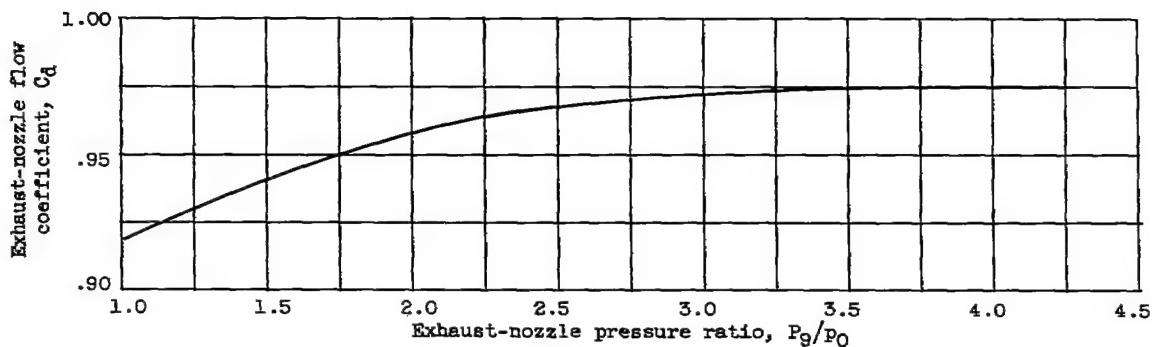
$$p_n = p_0$$

and in the choked case

$$P_n = \frac{P_9}{\left(\frac{1 + r_9}{2}\right)^{\frac{r_9 - 1}{r_9}}}$$

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The value of the flow coefficient was determined from reference 2 using the area ratio and cone angle of the particular nozzle employed in this investigation. The magnitude of the flow coefficient is presented in the following curve:



The compressor-inlet air flow was then determined from the nozzle gas flow by

$$W_{a,2} = W_{g,n} - W_{f,e} + W_{a,cl} \quad (4)$$

where the compressor leakage air flow $W_{a,cl}$ was measured at two instrumented mid-frame bleed ports.

The engine-inlet air flow $W_{a,1}$ based on pressure and temperature measurements in a bellmouth mounted on the front of the engine was determined by the same general equation as for the tail-pipe gas flow. The percentage of leakage at the section housing the inlet screens is

$$W_{a,1-2} = \frac{W_{a,1} - W_{a,2}}{W_{a,2}}$$

and was 3.3 percent of the compressor-inlet air flow $W_{a,2}$ for the range of conditions covered in this investigation.

Thrusts. - The jet thrust as determined from the thrust system measurements was calculated from the equation

$$F_{j,s} = F_d + (A_{seal} - A_g)(P_1 - P_{seal}) + A_g(P_1 - p_0) + 0.80 \left(\frac{1}{2} \frac{W_{a,1}}{g} V_{0a} \right) \quad (5)$$
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where the last term is the momentum force existing at the bellmouth inlet which was experimentally determined by instrumentation located on the surfaces of the bellmouth and bullet along with instrumentation at station 1. The net thrust will be determined by subtracting the equivalent momentum of the air at the engine inlet from the jet thrust.

$$F_{n,s} = F_{j,s} - \frac{W_{a,1} V_0}{g} = F_{j,s} - \frac{(W_{a,2} + W_{a,1-2})V_0}{g} \quad (6)$$

Jet thrust coefficient. - The jet thrust coefficient is defined as the ratio of scale jet thrust to rake jet thrust:

$$C_j = \frac{F_{j,s}}{F_{j,r}} \quad (7)$$

where

$$F_{j,r} = \frac{W_{g,n}}{g} V_n + A_n(p_n - p_0) \quad (8)$$

The charts in reference 3 were used in the solution of the preceding equation. When all the data obtained in this investigation were employed, the jet thrust coefficient was found to be independent of exhaust-nozzle pressure ratio and was a constant value of 0.99. The scatter in the coefficient values was approximately ± 1 percent for the range of conditions investigated.

Determination of performance for particular flight condition. - For a given flight condition, values of Re , δ , and θ can be obtained from table II. If these generalizing parameter values and engine speed are known, air flow, fuel flow, and exhaust-gas temperature can be obtained from the various performance curves. In order to determine

the net thrust, the jet thrust parameter must first be corrected to the desired flight condition to obtain the jet thrust. Then in order to obtain net thrust, the leakage between stations 1 and 2 must be added to the air flow for station 2 so that

$$F_n = F_j - \left(\frac{W_{a,2} + W_{a,1-2}}{g} \right) V_0$$

Sea-level static thrust ratings. - Because of the effect of inlet ram pressure on exhaust-gas temperature, data taken at an altitude of 5000 feet and flight Mach number of 0.2, which are included in the following table, had to be corrected to sea-level static conditions in order to determine the sea-level thrust for the engine.

Engine-inlet total pressure P_1 (lb/sq ft abs)	Engine-inlet total temperature T_1 (°R)	Nozzle-inlet total pressure P_g	Nozzle-inlet total temperature T_g (°R)	Engine manufacturer's 4-probe nozzle-inlet indicated temperature $T_{g,1}$ (°R)	Engine manufacturer's 5-probe nozzle-inlet indicated temperature $T_{g,2}$ (°R)	Corrected engine speed $N/\sqrt{\delta_1}$ (rpm)	Corrected compressor-inlet air flow $W_{a,2}\sqrt{\delta_1/\delta_1}$ (lb/sec)	Corrected compressor leakage air flow $W_{cl}\sqrt{\delta_1/\delta_1}$ (lb/sec)	Corrected compressor leakage air flow $\frac{W_{cl}}{\delta_1\sqrt{\delta_1}}$	Corrected engine fuel flow $W_{f,2}$ (lb/hr)
1812	537	3050	1568	1522	1519	7281	95.7	1.9	4681	
1814	537	3145	1612	1580	1560	7443	95.6	1.9	5008	
1813	534	3154	1601	1556	1553	7464	96.6	2.0	5014	
1812	537	3233	1656	1601	1604	7594	99.4	2.0	5348	
1816	537	3370	1728	1674	1676	7813	101.8	2.0	5870	
1813	537	3386	1731	1672	1680	7816	101.2	2.0	5916	
1814	533	3397	1736	1679	1683	7846	102.1	2.0	6022	

For sea-level static engine-inlet conditions, an engine speed of 7950 rpm, and a given exhaust-gas temperature, the tail-pipe total pressure may be determined from the engine-pumping-characteristic curves; therefore, the pressure ratio across the exhaust nozzle may also be determined. A plot of corrected fuel flow against engine temperature ratio will give the fuel flow for the proper exhaust-gas temperature. The compressor-inlet air flow may be determined from a plot of corrected air flow against corrected engine speed. In order to determine tail-pipe gas flow, compressor leakage air flow must be deducted and fuel flow added to the inlet air flow. From fuel flow, air flow, and exhaust-gas temperature, a value for γ_g may be obtained. All the factors that are required to calculate the rake jet thrust from equation (8) are now known. To the rake jet thrust there must be applied a jet thrust coefficient obtained from the value presented in this appendix in order to obtain the final sea-level jet thrust value.

The preceding sea-level static thrust calcualtion required the use of two assumptions:

- (1) The required nozzle-area change for the range of exhaust-gas temperatures of interest has no effect on the engine pumping characteristics.
- (2) The required nozzle-area change for the small change in exhaust-gas temperature has no effect on the curve of corrected air flow against corrected engine speed. Both of these assumptions were checked with data that were obtained during this investigation and verified as accurate and logical assumptions.

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TABLE I. - STANDARD

Reynolds number index Re	Engine speed N (rps)	Altitude static pressure P_0 (lb/sq ft abs)	Engine-inlet total pressure P_{in} (lb/sq ft abs)	Engine-inlet total temperature T_{in} (°R)	Baro-pressure ratio P_0/P_0'	Engine-inlet total tem- perature T_{in} (°R)	Compressor- outlet total pressure P_{c} (lb/sq ft abs)	Compressor- outlet total temperature T_c (°R)	Turbine-inlet total pres- sure, P_{t} (lb/sq ft abs)	Turbine-outlet total pres- sure, P_{t}' (lb/sq ft abs)	Turbine-outlet total tempera- ture, T_t (°R)	Nozzle-inlet ratio pres- sure, P_0/P_t (lb/sq ft abs)	Nozzle-inlet total temper- ature, T_n (°R)		
0.147	5953	205	232	415	902	681	860	345	1128	335	263	1102			
151	6362	208	258	1154	415	1058	695	1008	585	1234	376	292	1208		
156	7193	203	246	1209	415	1355	781	1267	490	1481	468	354	1477		
154	7409	201	241	1189	414	1358	780	1292	504	1570	481	372	1545		
151	7578	198	237	1205	414	1385	800	1319	516	1662	482	381	1638		
150	7707	192	236	1221	414	1212	810	1344	528	1722	504	390	1725		
202	5927	216	315	1149	412	2111	650	1152	443	1036	427	327	1007		
200	6362	215	313	1157	413	1003	684	1157	513	1164	491	378	1165		
201	6801	212	313	1175	412	1582	718	1201	577	1293	535	427	1305		
202	7207	211	316	1197	412	1713	755	1226	651	1442	504	466	1442		
200	7407	209	313	1199	412	1780	772	1292	655	1521	627	484	1531		
198	7574	206	314	1153	413	1828	791	1743	675	1808	646	504	1802		
198	7726	210	311	1162	414	1698	807	1807	702	1888	671	521	1884		
249	5921	241	392	1182	414	1885	647	1409	545	998	525	400	992		
248	6358	238	389	1164	414	1724	682	1644	632	1180	608	468	1138		
248	6815	237	389	1184	414	1958	718	1860	714	1270	588	527	1273		
250	7207	236	391	1162	413	2118	750	2010	777	1420	745	576	1418		
247	7409	236	389	1152	414	2189	770	2081	805	1490	772	597	1437		
249	7570	237	389	1159	413	2276	787	2187	837	1564	802	618	1571		
245	7818	239	386	1151	413	2349	814	2235	871	1688	834	648	1707		
298	5832	367	470	1280	414	1817	652	1727	675	1006	650	506	1011		
296	6558	354	468	1325	416	2091	684	1894	764	1117	735	566	1128		
302	6380	273	473	1172	413	2092	679	1895	764	1111	735	567	1122		
298	6815	351	468	1153	416	2335	719	2218	854	1263	819	631	1271		
311	6822	280	486	1178	412	2448	715	2324	890	1246	855	659	1268		
303	7193	277	473	1171	412	2561	745	2431	936	1378	889	698	1398		
287	7195	344	468	1160	415	2523	752	2395	926	1388	888	668	1400		
298	7407	348	466	1177	415	2822	770	2494	965	1480	926	716	1484		
300	7415	278	469	1186	412	2651	766	2521	974	1473	954	724	1488		
297	7570	338	465	1180	413	2699	786	2571	985	1542	954	737	1557		
303	7574	278	473	1172	412	2755	784	2624	1024	1553	881	768	1570		
302	7725	273	469	1181	411	2818	800	2685	1054	1630	1008	782	1652		
401	5923	553	740	1136	467	2481	699	2345	918	998	886	702	1003		
404	6350	555	745	1159	456	2913	733	2775	1072	1116	1030	800	1138		
401	6813	555	744	1140	459	2593	774	1194	1233	1263	1162	811	1294		
405	7189	547	759	1151	454	3698	800	3516	1356	1381	1302	1008	1415		
397	7405	553	757	1130	470	3834	824	3538	1403	1459	1410	1090	1555		
404	7570	554	747	1148	458	3973	840	3789	1468	1527	1410	1090	1555		
392	7247	543	741	1148	458	470	4258	878	4056	1580	1708	1521	1178	1732	
406	7856	552	745	1150	456	4885	874	4083	1580	1708	1523	1184	1745		
434	5930	508	718	1145	430	2618	655	2463	958	928	713	779			
431	6382	508	721	1140	434	3015	705	2659	1106	1100	1082	818	1118		
419	6817	505	719	1145	435	3731	745	5216	1240	1242	1189	918	1268		
431	7112	507	726	1148	436	3728	765	5359	1357	1335	1302	1006	1388		
425	7407	509	720	1148	438	3925	792	5728	1435	1444	1380	1065	1416		
418	7585	510	720	1142	433	3998	812	5801	1459	1512	1415	1095	1541		
428	7741	511	728	1144	437	4137	830	5837	1524	1591	1468	1138	1622		
418	7882	511	710	1151	436	4251	849	4052	1585	1702	1527	1189	1735		
510	5929	482	940	1151	467	3085	696	2907	1106	928	816	946			
508	6362	478	944	1177	469	3628	735	3442	1317	1081	1282	872	1106		
508	6817	480	939	1156	467	4201	770	4006	1357	1284	1175	1271			
508	7183	482	940	1151	468	4609	801	4387	1347	1624	1252	1388			
508	7409	474	939	1183	468	4750	818	4504	1778	1417	1699	1312	1488		
508	7565	510	720	1142	433	3998	812	5801	1459	1512	1415	1095	1541		
502	7718	477	831	1152	469	5075	851	4618	1882	1557	1784	1391	1602		
509	7722	485	839	1156	467	5138	848	4880	1887	1551	1818	1408			
503	7843	481	930	1155	468	5268	872	5011	1948	1668	1873	1457	1711		
505	7853	482	940	1151	471	5320	875	5062	1960	1666	1891	1467	1711		
610	5930	520	1127	2158	468	3663	697	5442	1311	916	1256	966	931		
813	6362	519	1121	2161	464	4552	730	4128	1582	1065	1514	1166	1097		
610	6813	520	1123	2159	467	5008	768	4773	1651	1220	1759	1353	1547		
605	7183	526	1124	2138	470	5492	802	5229	2045	1345	1855	1498			
608	7411	520	1118	2148	466	5764	816	5470	2107	1511	2031	1671	1455		
609	7578	524	1123	2142	467	5981	831	5675	2190	1477	2111	1635	1596		
602	7722	528	1125	2140	472	6122	852	5812	2215	1543	2167	1677	1594		
607	7849	530	1131	2134	471	6193	874	6107	2363	1548	2288	1775	1708		
596	7851	520	1114	2144	472	6302	875	5934	2320	1855	2240	1740	1712		
509	5929	1053	1789	1188	537	4915	760	4608	1788	924	1723	1568	954		
501	6372	1053	1788	1170	538	5584	801	5587	2134	1076	2047	1587	1084		
507	6817	1050	1785	1170	537	6394	841	6582	2623	1254	2417	1662	1265		
509	7187	1050	1790	1175	537	7732	874	7392	2815	1377	2731	2111	1418		
509	7409	1047	1789	1170	537	8153	892	7807	3013	1450	2867	2241	1198		
507	7589	1050	1785	1170	537	8449	805	8098	3126	1497	3008	2324	1548		
509	7727	1053	1788	1188	537	8765	918	8389	3243	1588	3125	1812	1612		
503	7951	1034	1779	1171	538	9155	874	8744	3385	1883	3261	2353	1701		

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ENGINE PERFORMANCE DATA

Air frame no. 4-probe nozzle-inlet total pressure, P_0 (lb/sq ft abs)	Air frame no. 5-probe nozzle-inlet total pressure, P_0 (lb/sq ft abs)	Engine no. 4-probe nozzle-inlet indicated temperature, $T_{0,1}$ (°R)	Engine no. 5-probe nozzle-inlet indicated temperature, $T_{0,2}$ (°R)	Compressor inlet air flow, $W_{0,2}$ (lb/sec)	Engine fuel flow, W_f , e (lb/hr)	Fuel-air ratio r/a	Net thrust F_n , (lb)	Corrected engine speed $N/\sqrt{f_1}$, (rps)	Corrected compressor-inlet air flow, $W_{0,2}/\sqrt{f_1}$, (lb/sec)	Corrected fuel flow $W_f, e/\sqrt{f_1}$, (lb/hr)	Corrected jet thrust parameter ($P_f^4 P_0 A_h / \sqrt{f_1}$), (lb)
540	536	1057	1058	10.5	374	0.0103	391	258	6655	83.7	3815
562	562	1188	1187	11.4	462	0.0115	519	357	7115	90.5	4536
471	477	1427	1446	12.3	696	0.0155	763	542	8042	99.1	6896
482	459	1512	1521	12.9	762	0.0188	801	596	8298	101.2	7464
473	438	1582	1590	12.9	822	0.0181	817	611	8487	103.1	8233
505	508	1668	1670	12.9	863	0.0195	888	677	8840	105.5	8931
437	432	981	986	14.2	456	0.0087	605	261	6550	85.1	3289
488	489	1124	1132	15.3	575	0.0105	781	437	7132	92.4	4357
560	566	1285	1274	16.4	750	0.0126	968	588	7651	98.9	5532
607	615	1595	1405	17.0	885	0.0147	1105	704	8084	101.6	6611
628	637	1471	1478	17.1	973	0.0161	1185	765	8311	105.1	7588
647	653	1551	1558	17.2	1059	0.0175	1227	818	8490	104.4	8067
672	676	1636	1639	17.4	1158	0.0189	1262	882	8852	105.6	8798
535	528	964	967	17.7	515	0.0052	829	371	6532	85.3	3121
615	617	1105	1111	19.3	667	0.0101	1049	581	7121	93.6	4166
692	699	1240	1244	20.5	883	0.0122	1246	713	7655	99.5	5374
747	758	1569	1375	21.1	1052	0.0142	1398	848	8079	102.1	6446
772	783	1446	1455	21.3	1161	0.0154	1553	919	8298	103.8	7085
801	813	1513	1524	21.6	1255	0.0165	1544	985	8456	105.0	7653
832	841	1640	1648	21.6	1402	0.0185	1625	1078	8784	105.2	8583
661	661	874	882	21.4	628	0.0083	849	445	5844	85.1	3167
744	748	1097	1104	23.2	820	0.0100	1087	628	7102	93.8	4142
743	747	1088	1094	23.5	814	0.0098	1281	638	7150	93.5	4082
826	835	1233	1239	24.4	1058	0.0120	1372	875	7612	98.9	5243
661	671	1227	1233	25.7	1074	0.0118	1578	871	7654	99.8	5247
501	514	1556	1560	25.7	1285	0.0139	1758	1044	8071	102.5	5346
680	692	1559	1562	25.5	1246	0.0140	1558	1011	8044	102.2	6294
925	939	1436	1444	25.6	1376	0.0152	1670	1122	8303	103.5	7008
855	848	1437	1447	25.9	1388	0.0152	1830	1141	8120	104.3	7053
954	958	1508	1514	25.8	1163	0.0181	1720	8486	104.5	7114	101.7
881	893	1512	1523	25.4	1521	0.0183	1982	1281	8428	105.3	7153
1008	1018	1592	1602	26.6	1617	0.0176	1973	1278	8983	106.7	8351
898	901	982	977	26.7	1666	0.0075	1067	454	6243	77.8	2509
1037	1049	1102	1107	32.2	1082	0.0095	1469	788	8710	87.1	5133
1188	1187	1265	1258	34.9	1395	0.0114	1858	1098	7187	94.3	4172
1305	1315	1371	1372	36.9	1708	0.0130	2185	1574	7617	99.8	5172
1348	1358	1450	1448	37.1	1872	0.0144	2246	1485	7785	101.7	5847
1409	1419	1508	1505	38.1	2043	0.0153	2447	1610	7971	102.7	6115
1518	1522	1679	1676	38.9	2480	0.0182	2705	1882	8352	105.8	7444
1523	1526	1698	1691	38.8	2490	0.0182	2743	1895	8391	104.1	7227
933	941	950	957	31.0	858	0.0076	1276	588	6117	83.7	2716
1066	1064	1084	1086	33.7	1116	0.0084	1631	856	6560	90.1	3582
1191	1208	1231	1231	35.6	1418	0.0113	1892	1164	7383	96.6	4516
1303	1318	1331	1329	37.6	1700	0.0128	2279	1412	7759	100.5	5418
1378	1394	1431	1431	38.3	1944	0.0144	2478	1610	8068	103.1	6219
1414	1427	1498	1498	38.5	2082	0.0154	2557	1689	8195	104.1	6331
1464	1479	1570	1575	38.9	2268	0.0168	2897	1798	8399	104.3	7153
1522	1533	1678	1682	39.1	2555	0.0186	2809	1955	8871	107.0	6294
1075	1083	917	917	37.0	858	0.0066	1702	508	6249	79.0	2036
1272	1286	1079	1080	40.8	1218	0.0087	2251	858	6893	88.8	2812
1478	1495	1240	1236	44.5	1708	0.0105	2779	1441	7185	95.2	3056
1625	1640	1382	1386	46.8	2098	0.0127	3169	1675	7574	100.0	4975
1693	1714	1424	1422	47.7	2317	0.0158	3385	1842	7872	102.1	5497
1758	1772	1491	1490	48.3	2613	0.0148	3520	1887	7553	103.0	5335
1793	1805	1555	1555	48.2	2661	0.0158	3618	2081	8399	104.3	7153
1815	1828	1550	1551	48.9	2722	0.0158	3644	2096	8159	104.6	6167
1871	1881	1659	1651	48.6	3015	0.0177	3819	2278	8564	105.1	7777
1869	1837	1658	1660	49.1	3032	0.0176	3861	2289	8351	105.3	7165
1275	1283	901	907	44.2	882	0.0063	2119	577	6214	78.8	1813
1525	1544	1069	1069	49.2	1455	0.0084	2786	1103	8125	81.9	2905
1760	1783	1231	1227	51.5	2008	0.0107	3448	1642	8168	83.5	3253
1939	1952	1357	1353	50.5	2480	0.0125	3878	1959	7569	101.1	4936
2028	2042	1422	1422	51.2	2763	0.0158	4127	2181	7818	102.6	5242
6412	2125	1485	1481	58.2	3005	0.0147	4549	2352	7987	104.0	5967
2164	2171	1546	1544	58.4	3217	0.0157	4472	2497	8100	104.8	6358
2280	2290	1659	1656	59.5	3656	0.0175	4775	2765	8346	106.1	7161
2236	2243	1663	1659	58.2	3593	0.0176	4637	2726	8341	105.4	7157
1748	1755	897	897	58.7	1102	0.0054	2180	528	5828	70.6	1382
2057	2076	1051	1058	60.1	1716	0.0074	3080	1010	6259	79.6	199
2409	2413	1250	1228	72.6	2536	0.0100	4016	1748	6702	87.5	2957
2723	2750	1580	1575	77.9	3323	0.0121	4839	2401	7075	93.7	3861
2882	2903	1454	1448	80.4	3765	0.0134	5289	2784	7284	96.7	4579
3084	3007	1510	1504	82.1	3086	0.0145	5535	3007	7410	98.9	4760
3115	3117	1585	1562	83.8	4406	0.0150	5878	3305	7595	100.5	5124
3262	3266	1648	1652	84.9	4953	0.0167	6252	3614	7809	102.8	5784

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TABLE II.- REYNOLDS NUMBER INDEX VARIATION WITH
FLIGHT MACH NUMBER AND ALTITUDE
[Ram-pressure recovery, 1.00.]

Altitude (ft)	Flight Mach number M_0	δ	θ	φ	Reynolds number index $\delta/\theta\sqrt{\theta}$	Altitude (ft)	Flight Mach number M_0	δ	θ	φ	Reynolds number index $\delta/\theta\sqrt{\theta}$
0	0	1.000	1.000	1.000	1.000	30,000	0.6	0.3787	0.8509	0.8862	0.4633
	.1	1.007	1.002	1.002	1.004		.7	.4118	.8715	.9029	.4886
	.2	1.028	1.008	1.006	1.018		.8	.4522	.8954	.9207	.5190
	.3	1.064	1.018	1.013	1.041		.9	.5019	.9222	.9416	.5551
	.4	1.117	1.032	1.023	1.075		1.0	.5619	.9524	.9655	.5964
	.5	1.186	1.050	1.036	1.117						
	.6	1.276	1.072	1.051	1.173						
	.7	1.387	1.098	1.069	1.238						
	.8	1.524	1.128	1.090	1.316						
	.9	1.691	1.182	1.117	1.404						
	1.0	1.893	1.200	1.141	1.516						
5,000	0	0.8318	0.9857	0.9753	0.8879	35,000	0	0.2352	0.7595	0.8149	0.3312
	.1	.8374	.9876	.9764	.8718		.1	.2368	.7611	.8184	.3325
	.2	.8554	.9734	.9809	.8839		.2	.2418	.7655	.8196	.3372
	.3	.8862	.9830	.9875	.9041		.3	.2502	.7732	.8257	.3446
	.4	.9291	.9965	.9973	.9333		.4	.2627	.7838	.8337	.3559
	.5	.9868	1.014	1.010	.9703		.5	.2789	.7975	.8443	.3699
	.6	1.061	1.035	1.025	1.018		.6	.3001	.8141	.8576	.3878
	.7	1.154	1.080	1.044	1.073		.7	.3262	.8359	.8727	.4093
	.8	1.268	1.089	1.064	1.141		.8	.3583	.8566	.8910	.4345
	.9	1.407	1.122	1.086	1.223		.9	.3977	.8825	.9111	.4647
	1.0	1.575	1.159	1.117	1.309		1.0	.4452	.9112	.9354	.4997
10,000	0	0.6881	0.9312	0.9491	0.7513	40,000	0	0.1853	0.7572	0.8130	0.2619
	.1	.6923	.9331	.9504	.7541		.1	.1866	.7588	.8141	.2631
	.2	.7075	.9387	.9549	.7647		.2	.1905	.7632	.8175	.2667
	.3	.7320	.9480	.9621	.7814		.3	.1972	.7709	.8239	.2726
	.4	.7684	.9609	.9714	.8069		.4	.2070	.7815	.8321	.2814
	.5	.8157	.9776	.9836	.8388		.5	.2198	.7950	.8430	.2924
	.6	.8778	.9983	.9989	.8794		.6	.2364	.8118	.8562	.3065
	.7	.9542	1.022	1.016	.9291		.7	.2570	.8314	.8714	.3235
	.8	1.048	1.050	1.037	.9859		.8	.2824	.8539	.8889	.3438
	.9	1.163	1.082	1.058	1.057		.9	.3134	.8798	.9090	.3676
	1.0	1.302	1.117	1.083	1.137		1.0	.3506	.9085	.9310	.3951
15,000	0	0.5643	0.8969	0.9223	0.6461	45,000	0	0.1459	0.7572	0.8130	0.2062
	.1	.5681	.8987	.9233	.6490		.1	.1469	.7588	.8141	.2071
	.2	.5799	.9040	.9281	.6572		.2	.1500	.7632	.8178	.2100
	.3	.6002	.9131	.9347	.6720		.3	.1552	.7709	.8239	.2145
	.4	.6300	.9256	.9448	.6931		.4	.1650	.7815	.8321	.2216
	.5	.6692	.9416	.9570	.7206		.5	.1730	.7950	.8430	.2302
	.6	.7198	.9615	.9719	.7553		.6	.1862	.8118	.8562	.2414
	.7	.7826	.9848	.9891	.7973		.7	.2024	.8314	.8714	.2548
	.8	.8601	1.012	1.008	.8482		.8	.2224	.8539	.8889	.2708
	.9	.9542	1.042	1.031	.9062		.9	.2467	.8798	.9090	.2894
	1.0	1.068	1.076	1.055	.9762		1.0	.2762	.9085	.9310	.3112
20,000	0	0.4586	0.8628	0.8980	0.5523	50,000	0	0.1149	0.7572	0.8130	0.1624
	.1	.4629	.8644	.8966	.5553		.1	.1157	.7588	.8141	.1631
	.2	.4726	.8696	.9016	.5622		.2	.1181	.7632	.8175	.1654
	.3	.4891	.8780	.9072	.5754		.3	.1223	.7709	.8239	.1691
	.4	.5132	.8902	.9172	.5930		.4	.1284	.7815	.8321	.1746
	.5	.5454	.9058	.9289	.6170		.5	.1362	.7950	.8430	.1812
	.6	.5865	.9247	.9440	.6461		.6	.1466	.8118	.8562	.1900
	.7	.6375	.9470	.9610	.6817		.7	.1594	.8314	.8714	.2006
	.8	.7004	.9728	.9798	.7248		.8	.1751	.8539	.8889	.2132
	.9	.7769	1.002	1.002	.7746		.9	.1943	.8798	.9090	.2279
	1.0	.8700	1.035	1.026	.8341		1.0	.2175	.9085	.9310	.2451
25,000	0	0.3710	0.8281	0.8682	0.4696	55,000	0	0.0905	0.7572	0.8130	0.1279
	.1	.3737	.8299	.8700	.4715		.1	.0911	.7588	.8141	.1285
	.2	.3814	.8347	.8740	.4778		.2	.0930	.7632	.8175	.1302
	.3	.3948	.8430	.8804	.4884		.3	.0963	.7709	.8239	.1331
	.4	.4145	.8545	.8891	.5043		.4	.1011	.7815	.8321	.1374
	.5	.4399	.8696	.9016	.5233		.5	.1073	.7950	.8430	.1428
	.6	.4731	.8877	.9151	.5487		.6	.1155	.8118	.8562	.1497
	.7	.5147	.9082	.9318	.5794		.7	.1255	.8314	.8714	.1580
	.8	.5657	.9338	.9515	.6152		.8	.1379	.8539	.8889	.1679
	.9	.6276	.9620	.9724	.6581		.9	.1530	.8798	.9090	.1795
	1.0	.7023	.9834	.9950	.7082		1.0	.1713	.9085	.9310	.1930
30,000	0	0.2968	0.7938	0.8414	0.3959	60,000	0	0.0713	0.7572	0.8130	0.1008
	.1	.2989	.7954	.8430	.3975		.1	.0717	.7588	.8141	.1011
	.2	.3052	.8002	.8469	.4029		.2	.0733	.7632	.8175	.1026
	.3	.3158	.8081	.8525	.4121		.3	.0758	.7709	.8239	.1048
	.4	.3315	.8183	.8621	.4248		.4	.0798	.7815	.8321	.1082
	.5	.3519	.8335	.8727	.4416		.5	.0845	.7950	.8430	.1124

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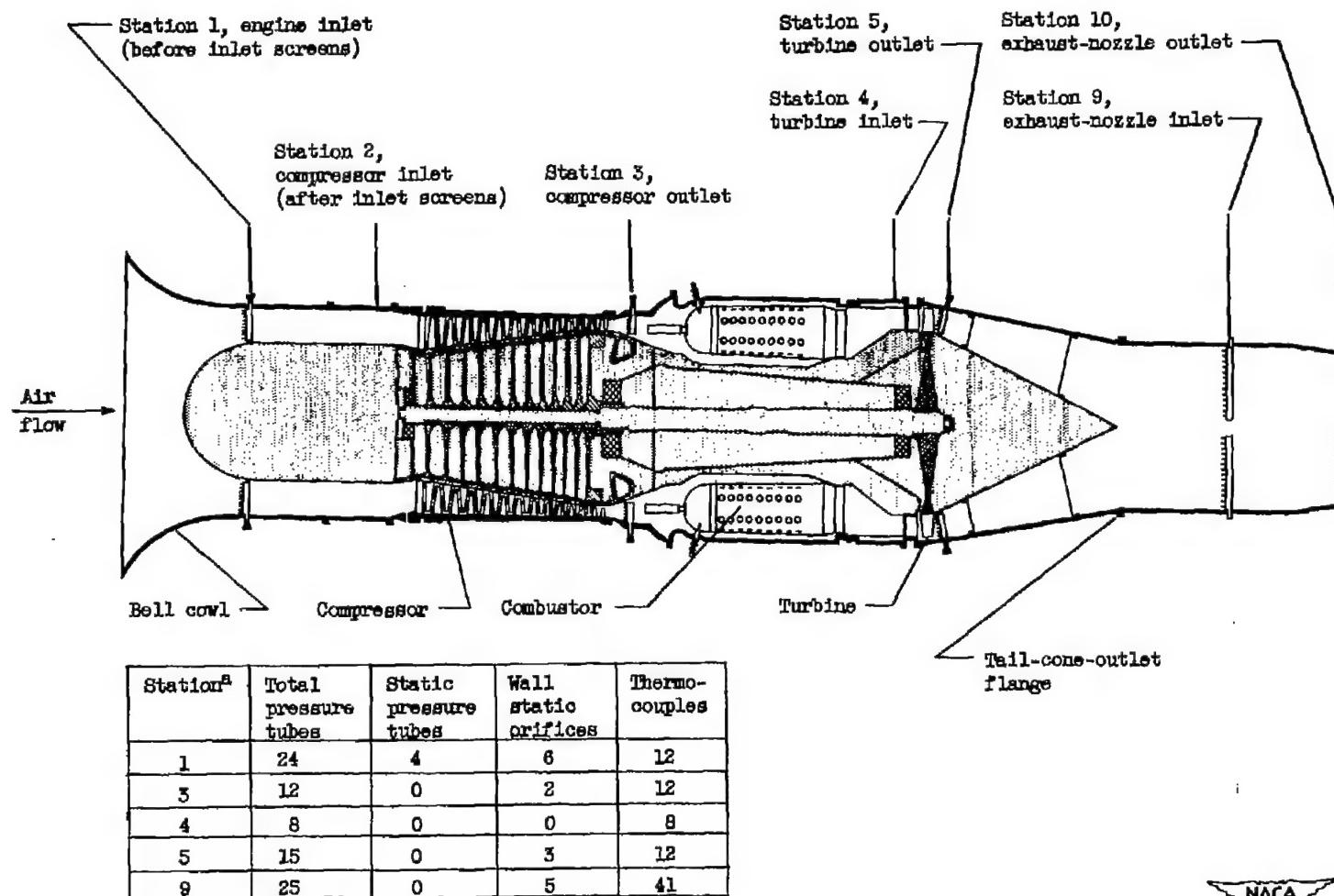
TABLE III. - PERFORMANCE DATA FOR EFFECT OF ENGINE-INLET TOTAL TEMPERATURE ON EXHAUST-GAS TOTAL TEMPERATURE

Engine speed N (rpm)	Altitude static pressure P ₀ (lb/sq ft abs)	Engine-inlet total pressure P ₁ (lb/sq ft abs)	Engine-inlet static pressure P ₁ (lb/sq ft abs)	Engine-inlet total temperature T ₁ (°R)	Nozzle-inlet total pressure P _g (lb/sq ft abs)	Nozzle-inlet total temperature T _g (°R)	Compressor-inlet air flow W _{a,2} (lb/sec)	Engine fuel flow W _{f,e} (lb/hr)	Net thrust F _n (lb)	Corrected engine speed N/√θ ₁ (rpm)	Corrected exhaust-gas total temperature T _{g/θ₁} (°R)
7947	967	996	894	431	2102	1690	54.9	3485	3063	8718	2035
7947	970	1002	898	455	2027	1678	53.1	3260	2881	8487	1915
7947	972	999	901	481	1955	1681	51.1	3084	2696	8257	1814
7951	968	999	902	499	1908	1897	49.6	2979	2572	8110	1765
7953	969	991	895	520	1869	1713	48.3	2911	2490	7945	1710
7953	969	995	800	532	1853	1731	47.7	2875	2422	7855	1689

TABLE IV. - PERFORMANCE DATA FOR EFFECT OF ENGINE-INLET RAM-PRESSURE RATIO ON CORRECTED EXHAUST-GAS TOTAL TEMPERATURE

Engine speed N (rpm)	Altitude static pressure P ₀ (lb/sq ft abs)	Engine-inlet total pressure P ₁ (lb/sq ft abs)	Engine-inlet static pressure P ₁ (lb/sq ft abs)	Engine-inlet total temperature T ₁ (°R)	Nozzle-inlet total pressure P _g (lb/sq ft abs)	Nozzle-inlet total temperature T _g (°R)	Compressor-inlet air flow W _{a,2} (lb/sec)	Engine fuel flow W _{f,e} (lb/hr)	Net thrust F _n (lb)	Corrected engine speed N/√θ ₁ (rpm)	Corrected exhaust-gas total temperature T _{g/θ₁} (°R)
7953	1294	1532	1204	512	2560	1741	65.3	4022	3460	8009	1764
7951	1223	1335	1207	513	2545	1731	65.3	3975	3229	7999	1752
7955	1175	1339	1211	512	2544	1729	65.4	3973	3176	8011	1753
7945	1128	1340	1211	511	2540	1719	65.7	3948	3092	8009	1747
7947	1042	1342	1213	512	2545	1718	66.0	3948	3040	8003	1742
7951	1290	1331	1207	528	2497	1758	63.3	3902	3307	7876	1725
7947	1220	1336	1212	528	2493	1743	65.6	3874	3077	7879	1713
7953	1169	1337	1211	529	2483	1741	63.5	3829	3010	7877	1708
7947	1120	1340	1214	529	2484	1738	63.7	3865	2985	7872	1705
7951	1083	1333	1207	529	2478	1732	63.8	3865	2922	7875	1699
7943	1047	1340	1212	529	2479	1728	64.0	3866	2888	7868	1695
7945	1455	1533	1394	536	2842	1743	71.7	4350	5562	7818	1888
7951	1464	1614	1487	537	2881	1727	75.7	4502	5569	7817	1669
7953	1471	1762	1597	536	3256	1726	83.1	4933	3790	7826	1671
7951	1468	1896	1713	537	3486	1712	89.7	5300	3987	7817	1655
7720	1784	1818	1855	531	3304	1669	84.0	4779	4228	7632	1631
7737	1726	1815	1855	538	3273	1673	82.2	4710	4086	7613	1620
7722	1691	1815	1858	535	3250	1657	82.0	4638	4001	7805	1607
7727	1597	1819	1858	534	3247	1653	82.3	4628	3922	7617	1607
7724	1516	1824	1859	534	3247	1642	82.8	4618	3832	7614	1598
7727	1441	1826	1859	533	3256	1642	83.3	4648	3800	7825	1599

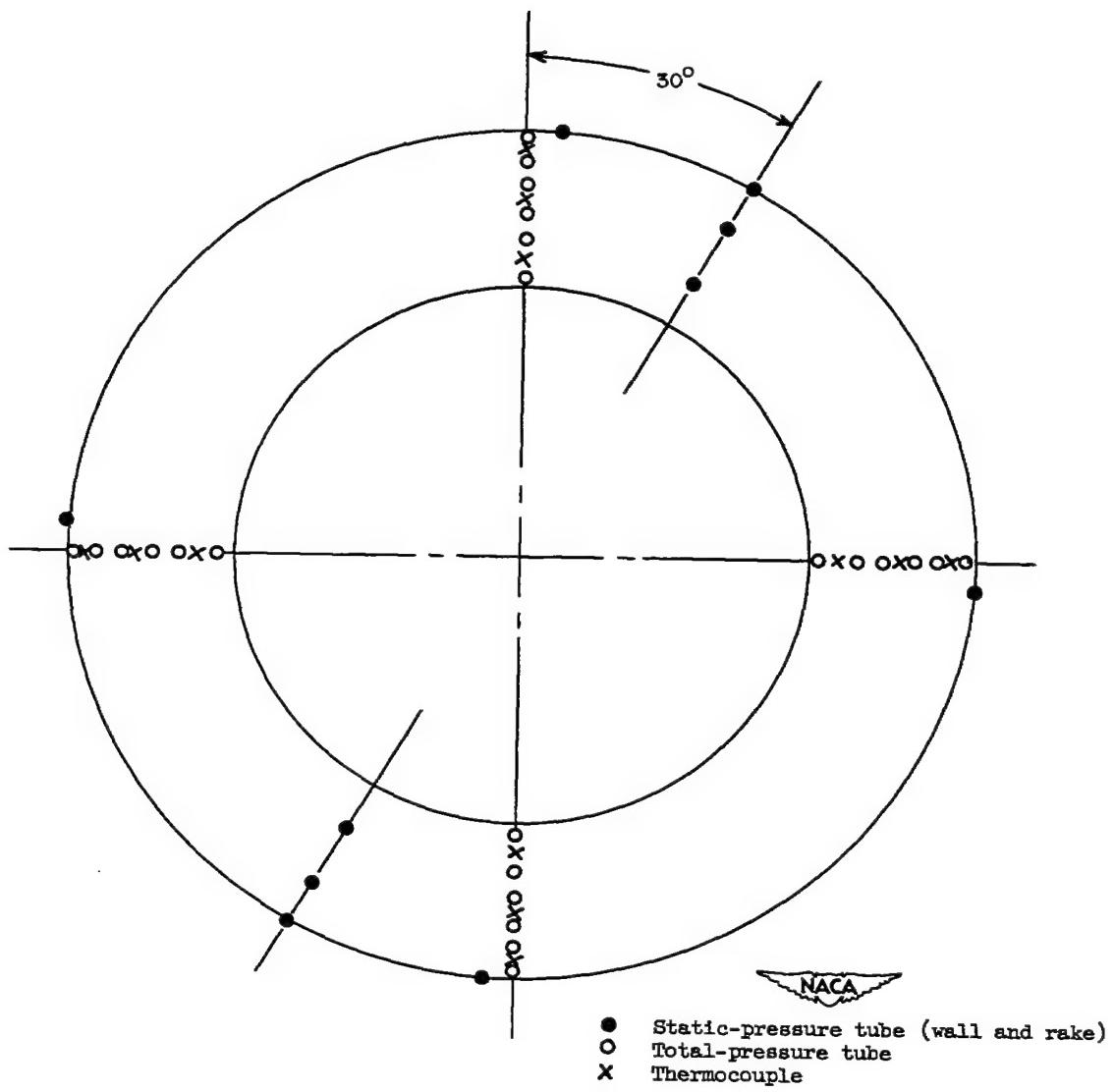
NACA



^aNo instrumentation at station 2.

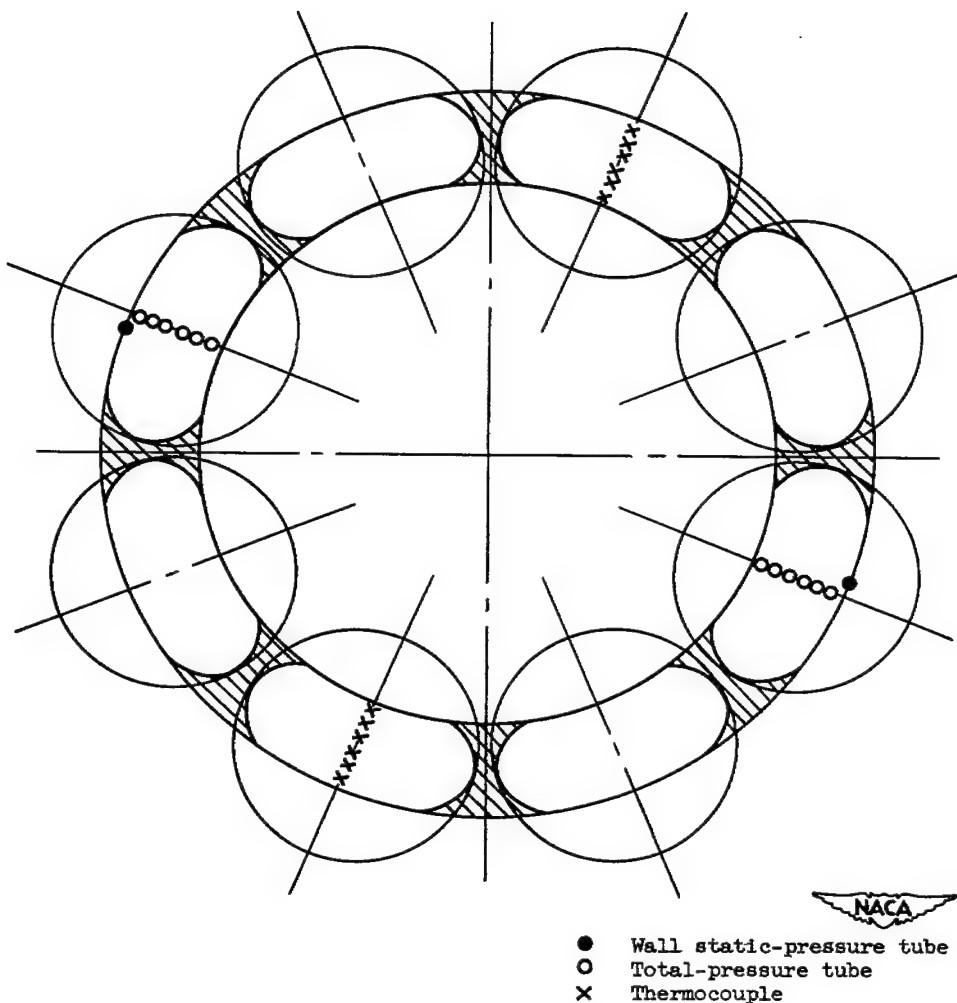


Figure 1. - Cross section of engine showing location of instrumentation.



(a) Instrumentation at engine inlet, station 1, 21 inches upstream of leading edge of compressor-inlet guide vanes. Viewed from upstream.

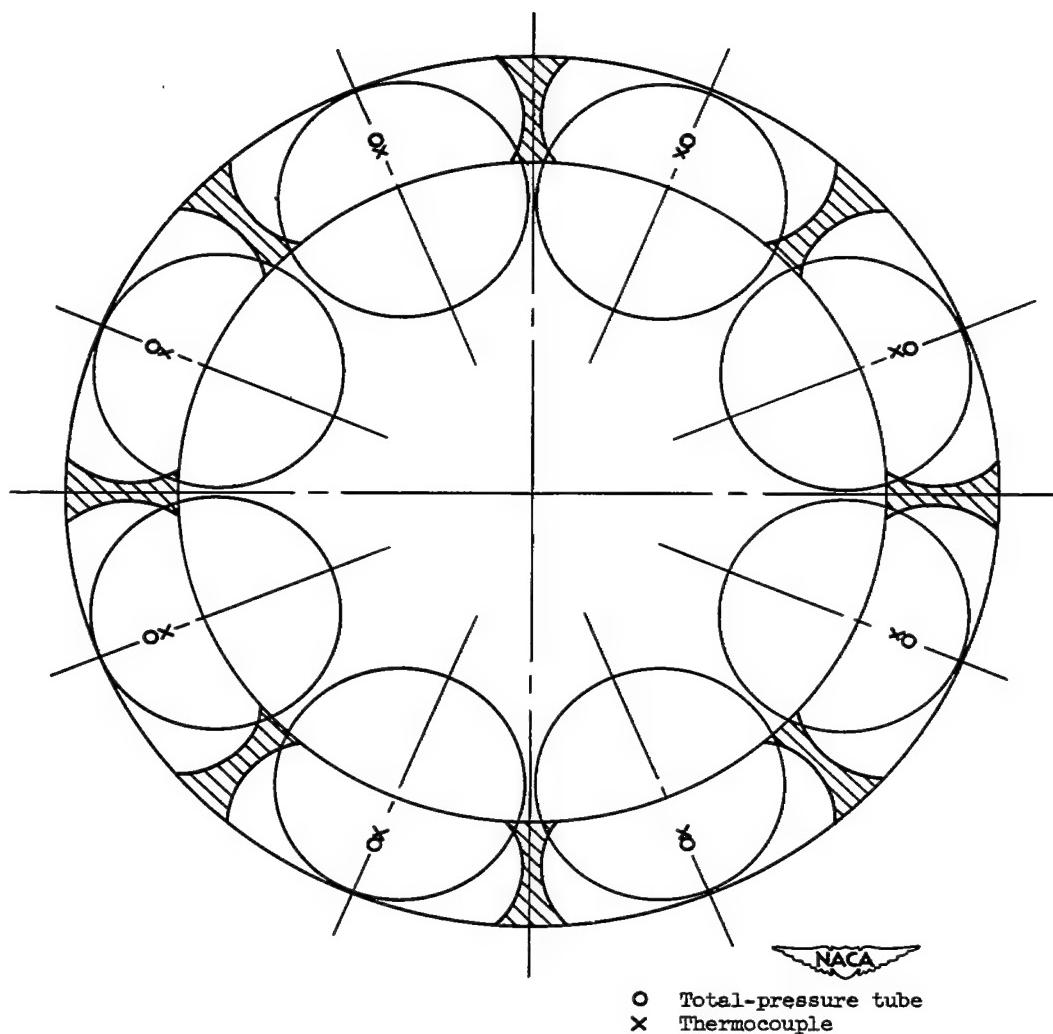
Figure 2. - Instrumentation sketches of various measuring stations.



(b) Instrumentation at compressor outlet, station 3, 2 inches downstream of trailing edge of compressor-outlet guide vanes. Viewed from upstream.

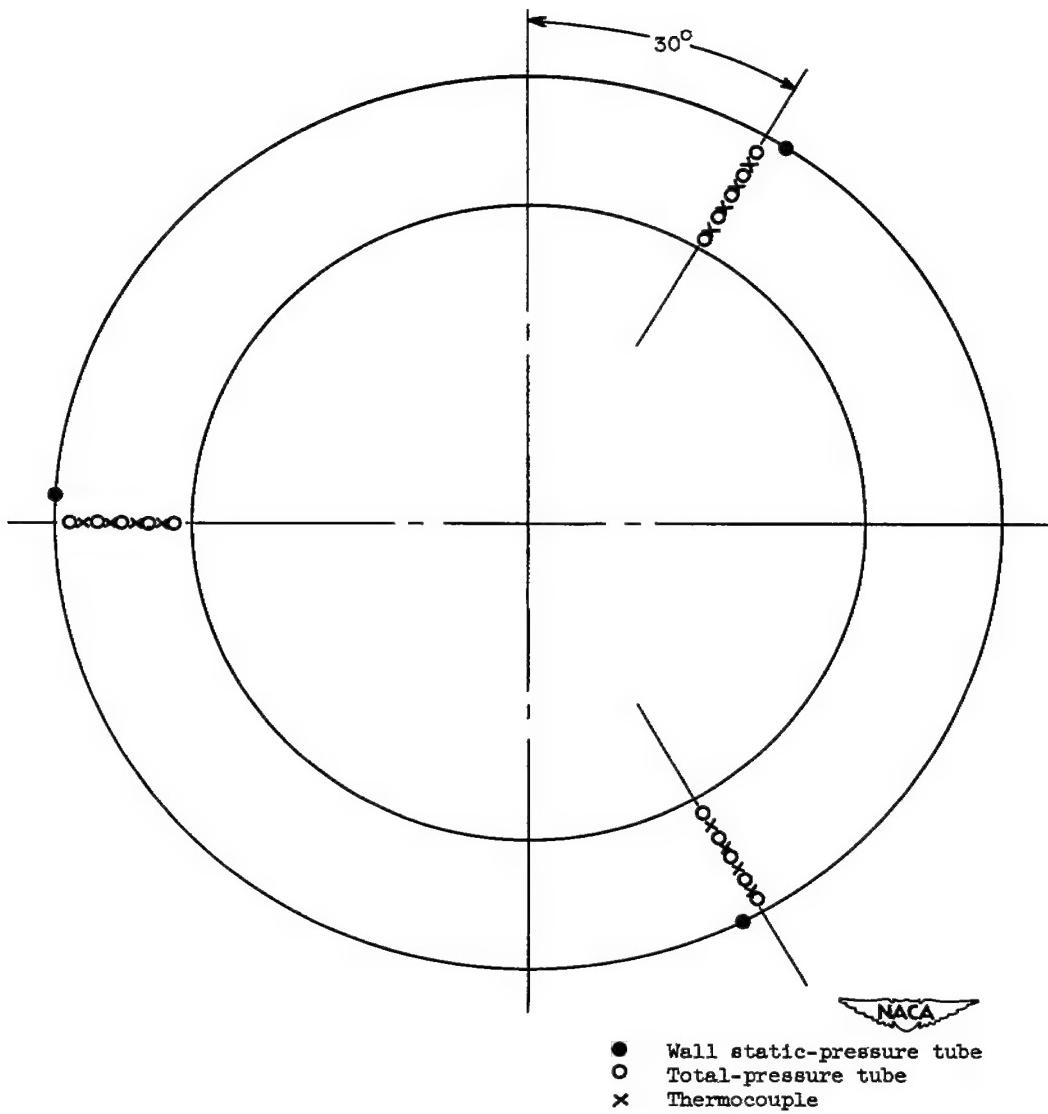
Figure 2. - Continued. Instrumentation sketches of various measuring stations.

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(c) Instrumentation at turbine inlet, station 4, $1\frac{3}{4}$ inches upstream of leading edge of turbine-inlet guide vanes. Viewed from upstream.

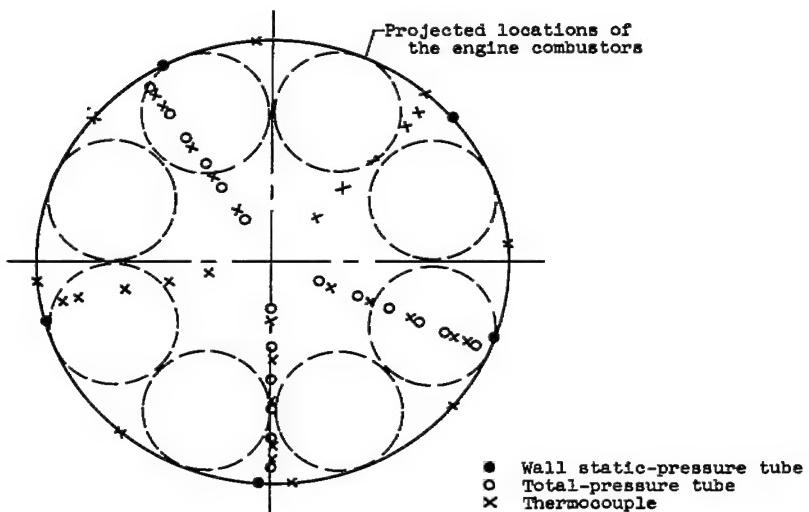
Figure 2. - Continued. Instrumentation sketches of various measuring stations.



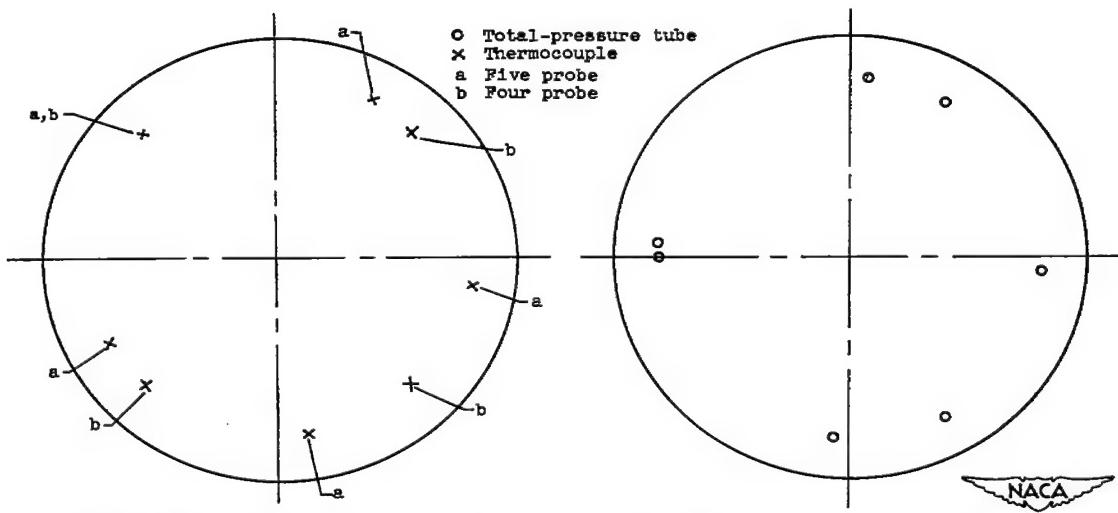
(d) Instrumentation at turbine outlet, station 5, $2\frac{3}{4}$ inches downstream of trailing edge of turbine blades. Viewed from upstream.

Figure 2. - Continued. Instrumentation sketches of various measuring stations.

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(e) NACA instrumentation at nozzle inlet, station 9, 15.15 inches downstream of tail-cone-outlet flange. Viewed from upstream.



(f) Engine and air frame manufacturers' instrumentation at nozzle inlet, station 9, 15.15 inches downstream of tail-cone-outlet flange. Viewed from upstream.

Figure 2. - Concluded. Instrumentation sketches of various measuring stations.

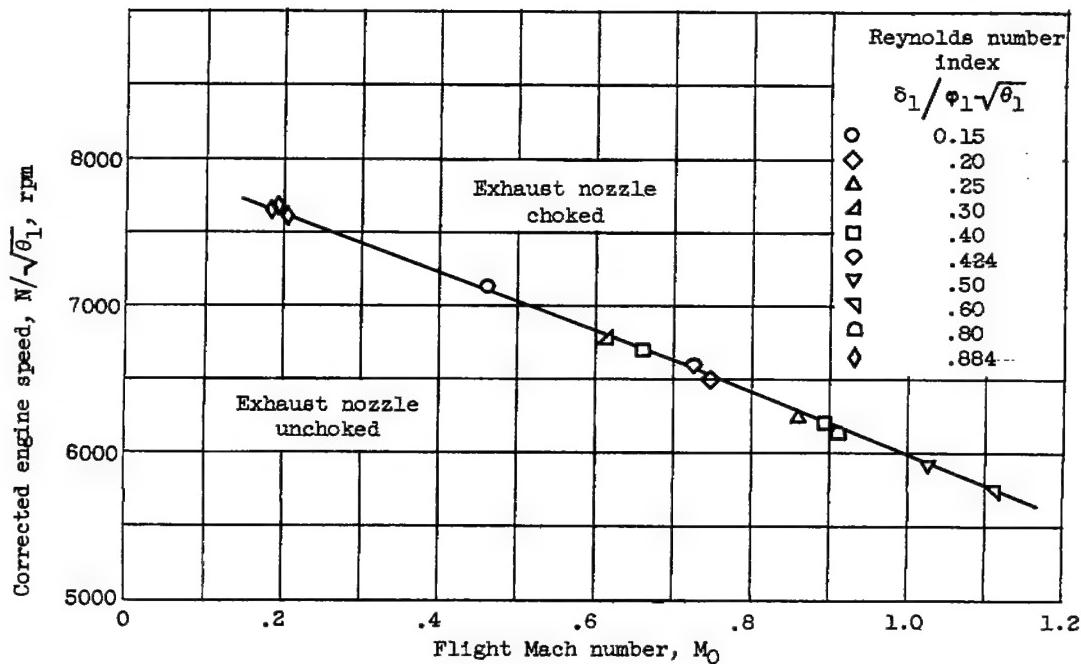


Figure 3. - Minimum corrected engine speeds at which critical flow existed in the exhaust nozzle.

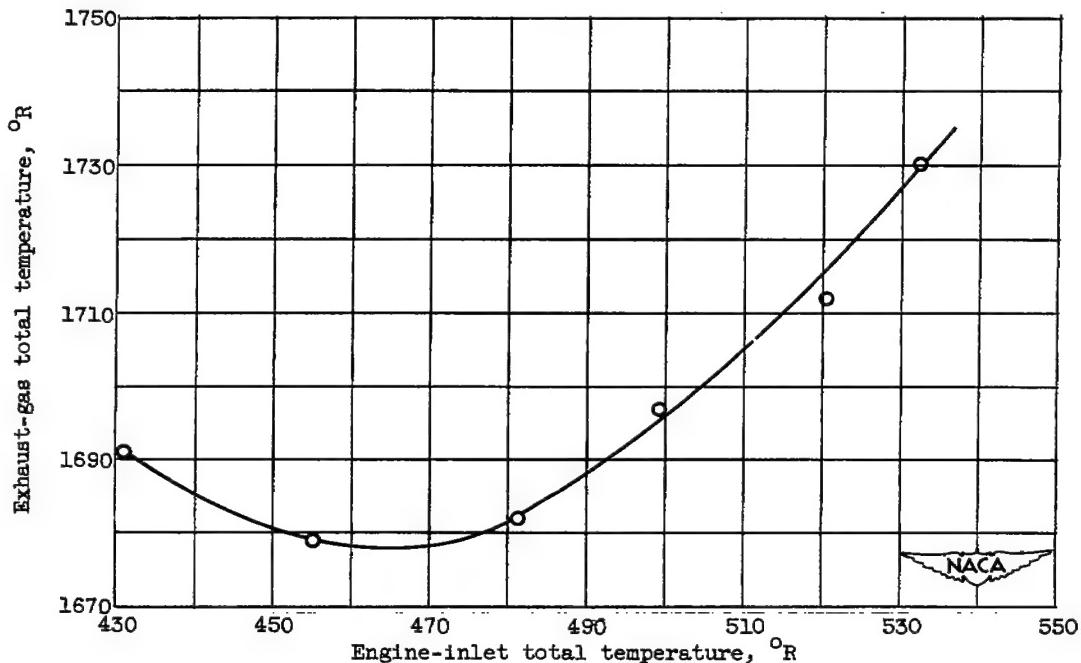


Figure 4. - Effect of engine-inlet total temperature on exhaust-gas total temperature. Engine speed, 7950 rpm; altitude, 20,000 feet; flight Mach number, 0.2.

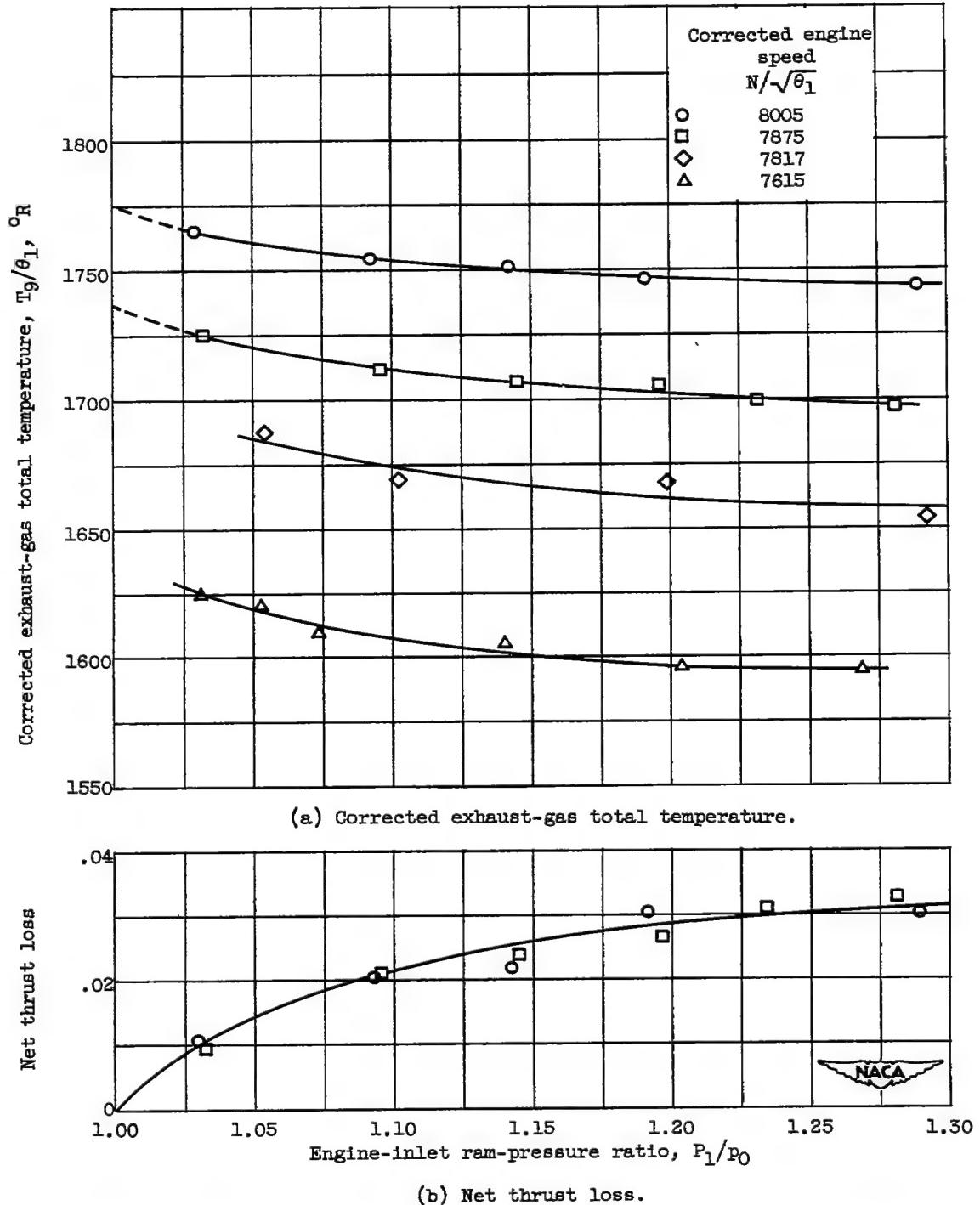


Figure 5. - Effect of engine-inlet ram-pressure ratio on corrected exhaust-gas total temperature and net thrust loss for various corrected engine speeds.

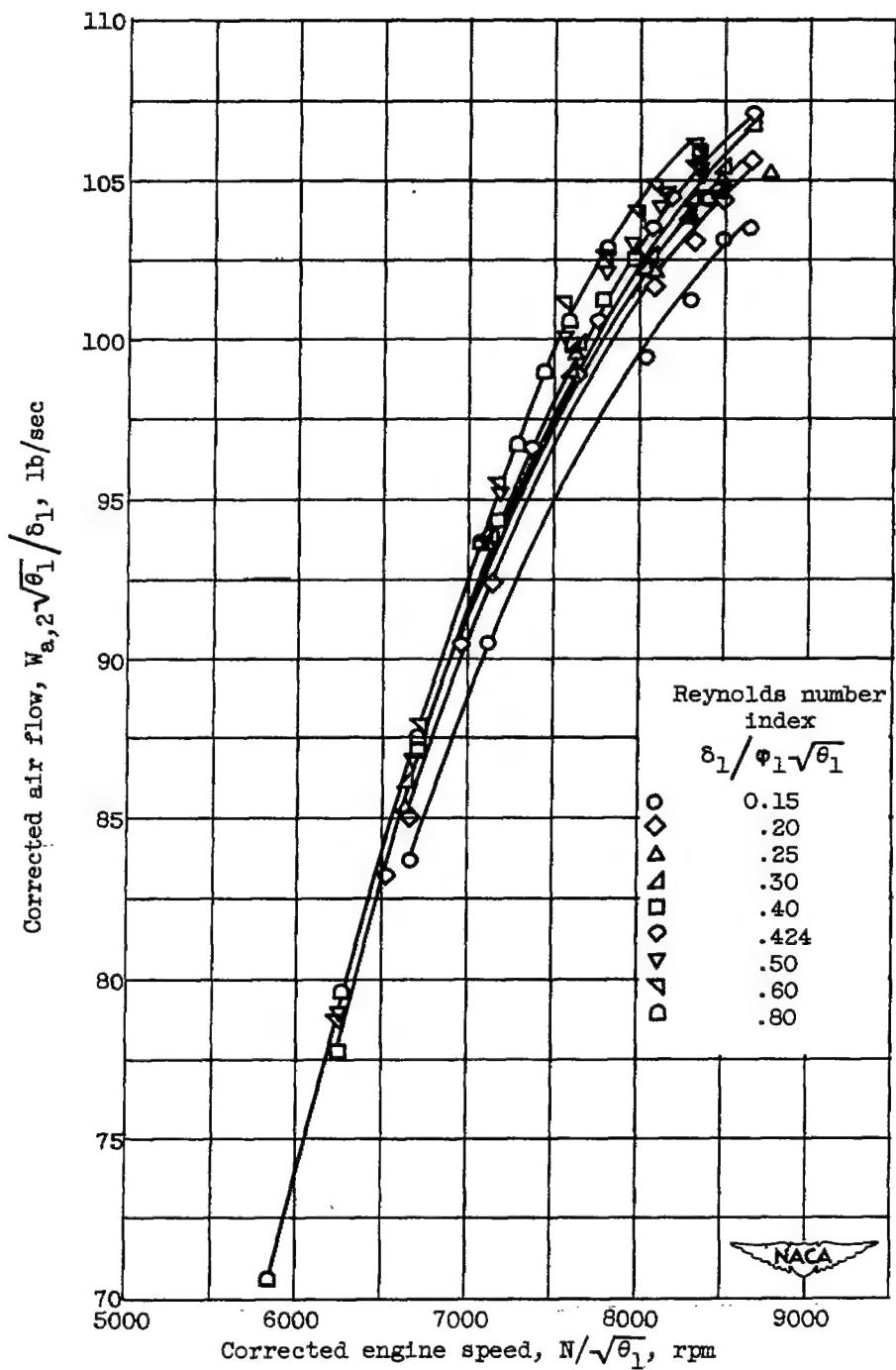


Figure 6. - Variation of corrected air flow with corrected engine speed for various Reynolds number indices.

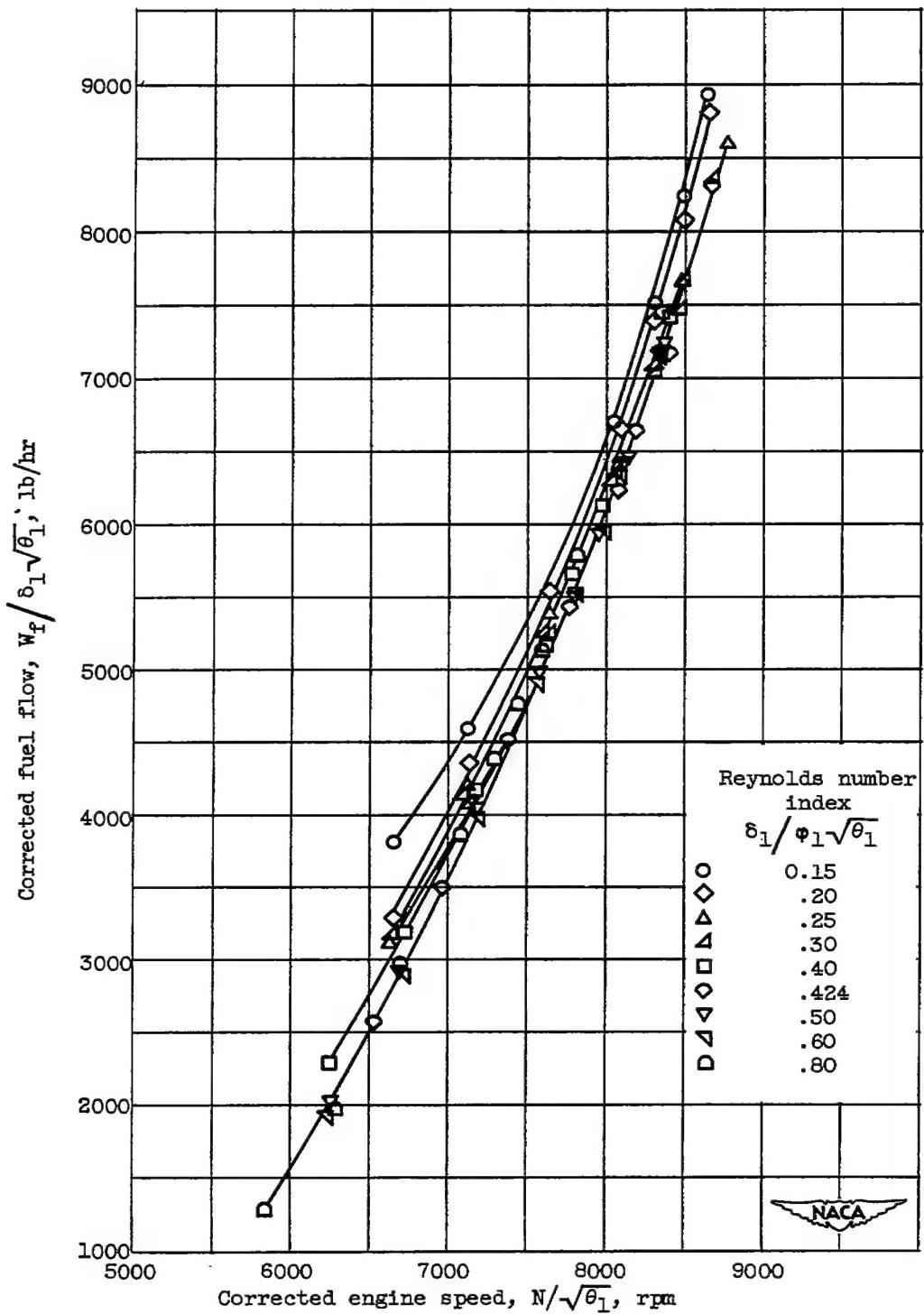


Figure 7. - Variation of corrected fuel flow with corrected engine speed for various Reynolds number indices.

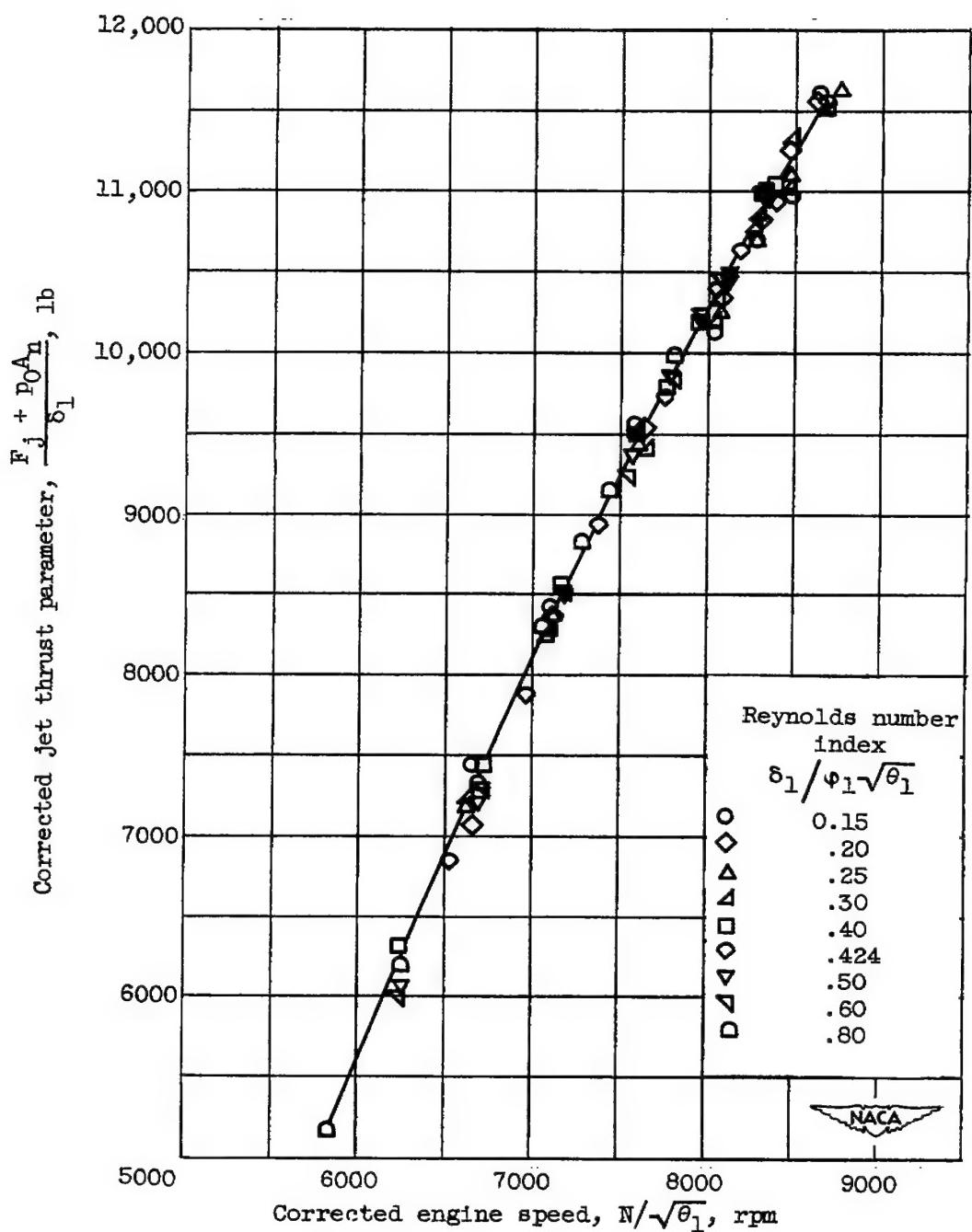


Figure 8. - Variation of corrected jet thrust parameter with corrected engine speed for various Reynolds number indices.

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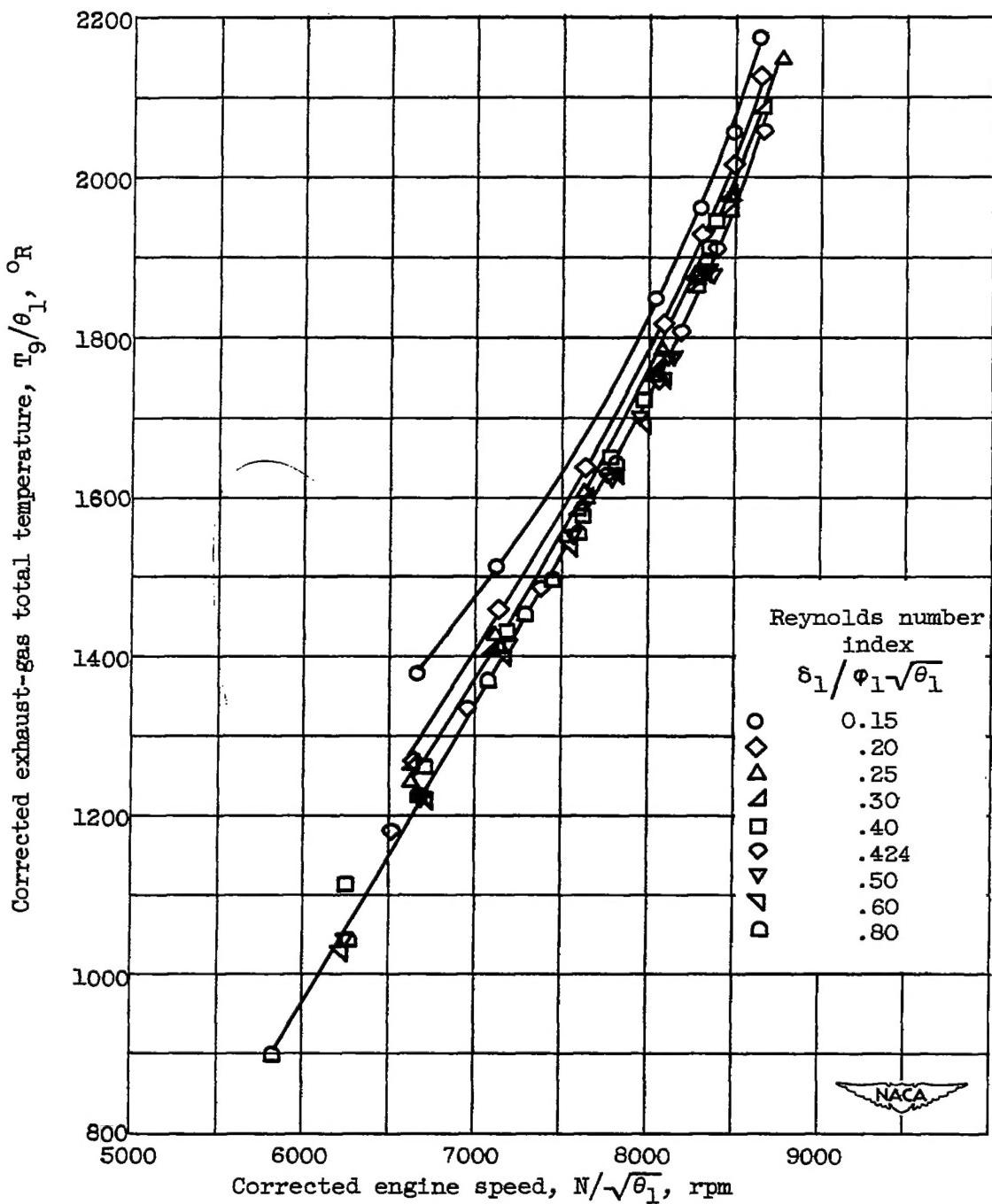


Figure 9. - Variation of corrected exhaust-gas total temperature with corrected engine speed for various Reynolds number indices.

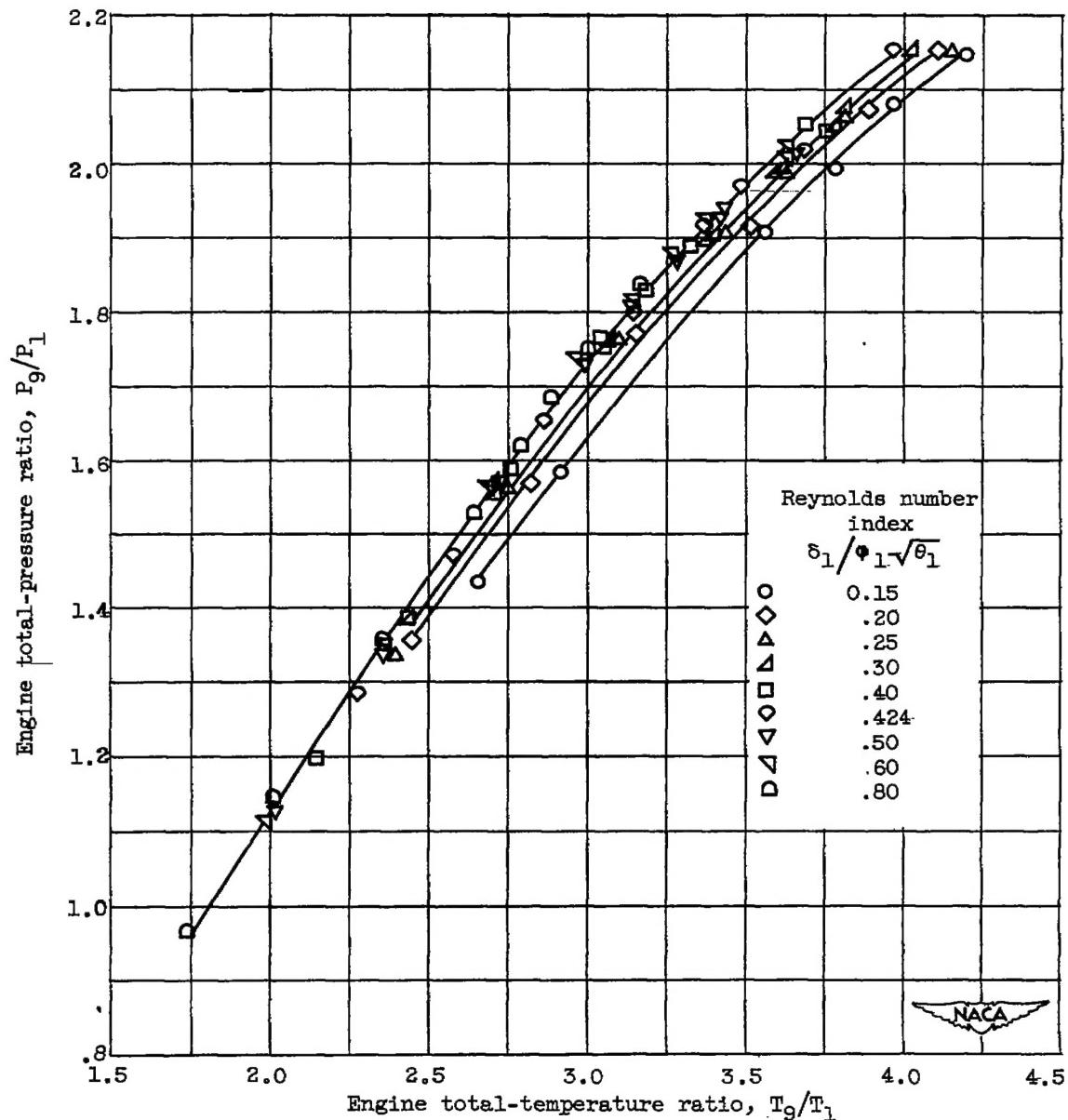


Figure 10. - Variation of engine total-pressure ratio with engine total-temperature ratio for various Reynolds number indices.

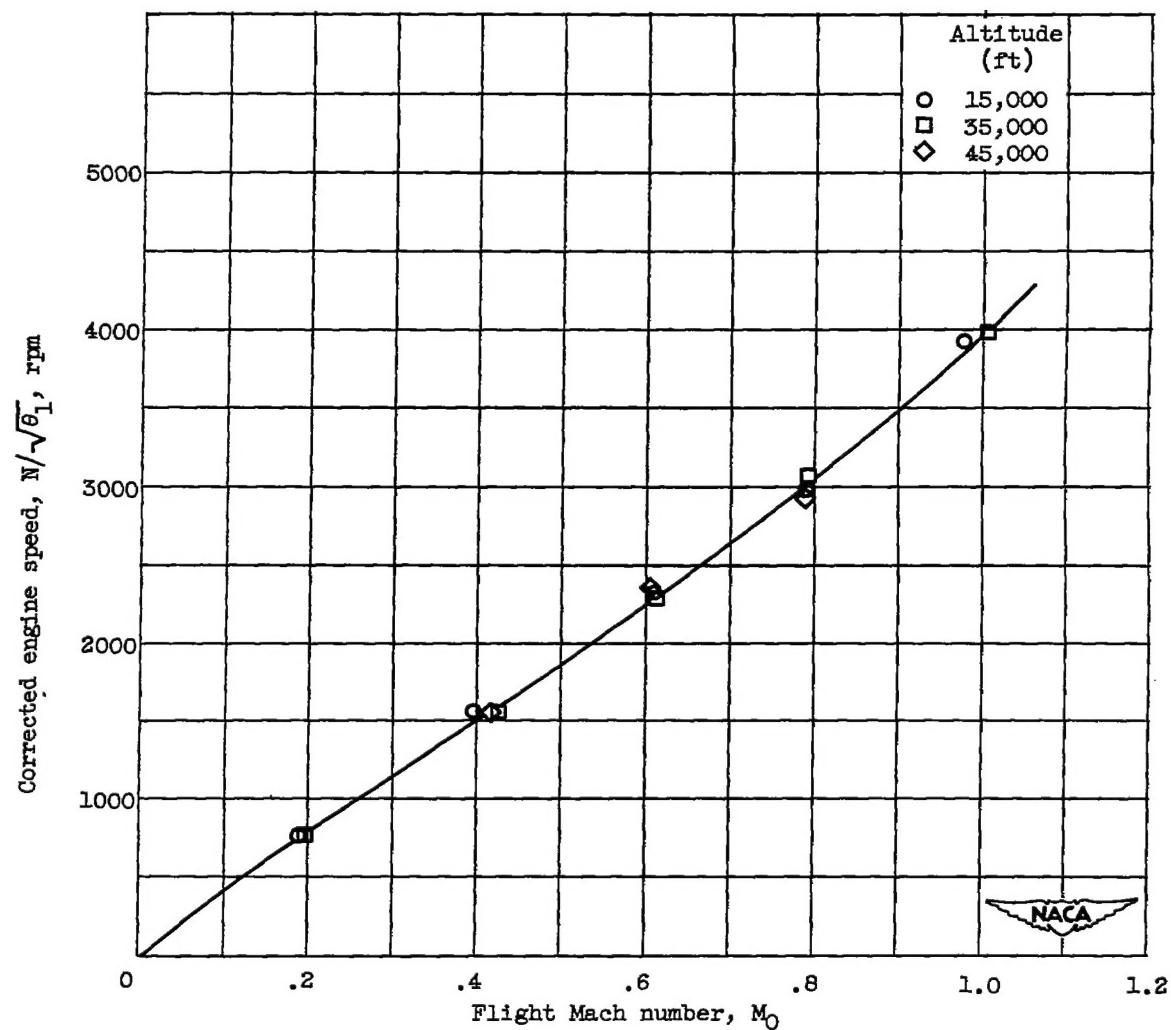


Figure 11. - Variation of corrected windmilling engine speed with flight Mach number at three altitudes.

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